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Durability patch: application of passive damping to high cycle fatigue cracking on aircraft

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ABSTRACT

Although high-cycle fatigue cracks in secondary structure are often termed "nuisance cracks", they are costly to repair. Often the repairs do not last long because the repaired part still responds in a resonant fashion to the environment. Although the use of visco-elastic materials for passive damping applications is well understood, there have been few applications to high-cycle fatigue problems because the design information: temperature, resonant response frequency, and strain levels are difficult to determine. The Damage Dosimeter, and the Durability Patch are an effort to resolve these problems with the application of compact, off-the-shelf electronics, and a damped bonded repair patch. This paper will present the electronics, and patch design concepts as well as damping performance test data from a laboratory patch demonstration experiment.

Key words: Passive Damping, High-Cycle Fatigue, Bonded Repair, Cocuring, co-curing, viscoelastic material, composite material, finite element analysis, damping, modal strain energy.

1 SUMMARY

The Durability Patch Program addresses the restoration of structural integrity of cracked secondary structure induced by resonant high cycle fatigue. The program is based on adapting technology from three basic areas:

- bonded structural repair,
- vibration damping, and
- avionics.

These three areas each possess a large technology base and have achieved a threshold of maturity sufficient to support this program. A typical repair would be for a crack less than four inches long in 0.050 inch thick skin of the upper trailing edge of a wing. Nuisance cracking is a high maintenance and repair cost item. Typical sources of excitation are: pressure pulses from engine 1st stage compressor, jet engine exhaust, disturbed air flow behind stores, separated flow on upper wing, air flow around open cavities, propeller tip vortices, etc. Typical locations of nuisance cracking are: flap skins, spoiler skins, rudder skins, aileron skins, weapon bay doors, wing trailing edges, etc. Of course there are other possible causes of cracking in secondary or lightly loaded structure besides resonant high cycle fatigue.

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2 HIGH CYCLE FATIGUE

High cycle fatigue life and crack growth rates are key disciplines in evaluating the longevity of structural repair. Methodology for calculation of resonant high cycle fatigue (HCF, sometimes called sonic fatigue or acoustic fatigue) life and associated crack growth rates used here is well established and consistent with standard industry practice.[1,2,3,4,5] It has been found that, in most cases, the HCF damage is due to linear resonant response in a single vibration mode; this implies that the vibratory stress is a narrow band random process. The threshold for number of cycles for high cycle fatigue is 10e6 (1,000,000) cycles. Fatigue consists of crack initiation, propagation and final rupture. The termination of the crack initiation phase is somewhat arbitrary since it depends on what is detectable and on what is acceptable in service. The basis for fatigue calculations is the S-N curve

$$S_{RMS} = S_{UHCF} N^b ; N = \left(S_{RMS}/S_{UHCF}\right)^{1/b} \tag{1}$$

where S_{RMS} is the rms stress, the coefficient S_{UHCF} may be considered to be a hypothetical ultimate rms high cycle fatigue stress which would cause failure at the first cycle, N is the fatigue life in number of cycles, and b is the Basquin parameter or exponent. This equation is a straight line when log-log scales are used for stress as a function of life. For 2024 aluminum the value of the exponent (Basquin parameter) is 0.1772 and the value of the coefficient is (98.26 ksi); for these values, a stress improvement factor (SIF) of 2 results in a life improvement factor (LIF) of 50, and fatigue strengths of (8.5 and 2.5 ksi) at 10e6 and 10e9 cycles respectively. Other aluminum alloys are not much different. Since HCF begins at 10e6 cycles, the upper threshold of interest for most aluminums is (8.5 ksi) rms, or a strain of 850 micro strain rms. This corresponds to approximately 3000 micro strain peak. Because of stress concentrations, uncertainties in locating strain gages, and averaging effects, measured strains are somewhat less.

It is envisioned that because of the existence of a crack, the life is known and the baseline or unrepaired stress level may be calculated; one objective is to reduce the stress level such that life is enhanced. Stress levels will be reduced through beef-up and through vibration damping using viscoelastic materials(vem's). The rms stress level is approximately proportional to the square root of modal damping; consequently, damping is a very useful approach for significant vibratory stress reduction. It happens that modal damping is dependent on the dynamic mechanical properties of the vem's, which in turn are dependent on service temperature and vibration frequency. It is therefore necessary to determine the vibration frequency and temperature at which damage accumulates in service. It is assumed that the temperature and stress time histories are available at the location of chronic nuisance cracking.

The analysis is performed for the i-th frequency band, the j-th temperature band, and the k-th time increment. If Φ is the Power Spectral Density (PSD) of stress, the rms stress is given by the square root of the area under the PSD curve.

$$S_{RMS} = \left[\int_{f_l}^{f_h} \Phi\left(f\right) df\right]^{1/2} \tag{2}$$

It may be desirable to calculate the contribution of one third octave (or other) bands

$$\varphi_{ik} = \left[\int_{f_{li}}^{f_{hi}} \Phi\left(f\right) df\right]^{1/2} \tag{3}$$

In this case, the rms is the square root of the sum of squares, but, because the response is dominated by only one vibration mode, it may be represented by any of the following, including the sum or a single term

$$S_{RMS} = \left[\sum_{i=1}^{N_i} \varphi_{ik}^2\right]^{1/2} \cong \sum_{i=1}^{N_i} \varphi_{ik} \cong \varphi_{ik_{\max}} \tag{4}$$

By substituting this expression for rms stress into the S-N curve, the fatigue life corresponding to that stress level may be found

$$N_{ijk} \cong (\varphi_{ik}/S_{UHCF})^{1/b} \tag{5}$$

Cumulative damage from different stress levels or time increments may be calculated by using the Palmgren-Miner rule (see Rudder[1] p. 195)

$$d = \sum n/N \tag{6}$$

where N is the number of cycles to failure at the stress level S; n is the number of cycles actually experienced at stress level S, (n/N) is the damage due to the n cycles; and d is the cumulative damage; when $d = 1$ a fatigue failure is indicated.

Time histories of the temperature and the one third octave bands are recorded

$$T_k, \varphi_{i,k}; i = 1, \ldots, N_i; k = 1, \ldots, N_k \tag{7}$$

A function may be defined as unity or zero based on whether or not the temperature for the k-th time increment is within the j-th temperature band

$$\delta_{T_k,T_j} = \{ \begin{matrix} 1; \text{if } T_k \text{ in } T_j \text{ band} \\ 0; \text{otherwise} \end{matrix} \tag{8}$$

The cumulative damage is given by

$$d_{ij} = \sum_{k=1}^{N_k} n_{ijk}/N_{ijk} \tag{9}$$

The number of cycles is

$$n_{ijk} = f_i \Delta t_k \tag{10}$$

The fatigue life at this stress level would be

$$N_{ijk} = (\varphi_{ik}/S_{UHCF})^{1/b} \delta_{T_k,T_j} \tag{11}$$

with appropriate substitution the cumulative damage may be calculated as a function of vibrational frequency and temperature

$$d_{ij} = \sum_{k=1}^{N_k} f_i \Delta t_k (\varphi_{ik}/S_{UHCF})^{1/b} \delta_{T_k,T_j} = d_{ij}(f_i, \Delta T_j) \tag{12}$$

and the capability to obtain this from service is crucial to the success of this program. The Dosimeter described below addresses this requirement. The above is the cumulative fatigue damage algorithm; it is adapted from standard industry practice and is judged sufficiently accurate for present purposes.

The acoustic noise excitation is typically represented by a broad band random uniform pressure field having a spectral density

$$G_p(f_1) \tag{13}$$

at the fundamental resonance frequency of the skin panel. The spectral density of the response is integrated over the frequency domain to obtain the expression for the mean square stress (see Rudder[1] p194)

$$\overline{\sigma^2(t)} = \frac{\pi f_1 G_p(f_1)}{2\eta} \left(\frac{\sigma_0}{P_0}\right)^2 \tag{14}$$

where σ_0 is the static stress at the appropriate location produced by the uniformly distributed pressure P_0 and and η is the modal damping. (Some investigators use the fraction of critical viscous damping ratio.)

Substitution leads to an expression for the ratio of repaired and unrepaired (ie, baseline) fatigue lives

$$\left[\frac{N_R}{N_U}\right] = \left[\left(\frac{\eta_U}{\eta_R}\right) * \left(\frac{f_R}{f_U}\right) * \left(\frac{\sigma_{OR}}{\sigma_{OU}}\right)^2\right]^{1/2b} \tag{15}$$

The above apply to total fatigue life; it is also desired to quantify unrepaired and repaired crack growth rates of existing cracks. The quantity of primary importance which influences the growth rate in typical aircraft skin structural materials of a fatigue crack is the variation of the crack tip stress intensity factor and the Paris equation is used

$$\frac{da}{dN} = C \Delta K_{RMS}^n \tag{16}$$

The parameters C and n are material properties. Most of the work has been done for centrally cracked thin sheets subjected to cyclically varying inplane loads, whereas here the interest is in edge cracked panels subjected to flexural loading or bending. Byrne[2] has arrived at the expression

$$K = 0.8\sigma_\infty a^{1/2} \tag{17}$$

for an edge cracked semi-infinite plate deformed into a cylindrical shape(ie, cylindrically bent). Different methods to calculate stress intensity, including FEA, will be considered.

3 SURVEY

In the interests of determining the nature and extent of maintenance and repair costs as a result of nuisance cracking, 126 copies of a survey letter were sent to structural sheet metal shops on flight lines. Additionally, expert personnel visited four flight lines to further assess available facilities, equipment, and personnel skills. It was concluded that this type of maintenance and repair is not accurately and completely documented. Also, typically, the logistics structures engineers are not fully aware of the nature and extent of nuisance cracking.

It was learned that almost all cracks are discovered and repaired before they reach a length of 4 inches. Also, scheduled flying and alert status dominates maintenance and repair techniques. A wing commander would be reluctant to accept a new repair technique if it required significantly more manhours or clock time to implement. A typical small non-flush mechanically fastened sheet metal patch requires two manhours to complete. This is accepted as a target for the present Durability Patch effort.

4 DOSIMETER

The Dosimeter has been conceived to gather service environmental data with regard to suspected resonant hcf cracks as economically as practical. Dosimeter requirements are that service data be gathered, processed and stored to permit

1. the design of a damping treatment (which requires the knowledge of the vibration frequency and temperature at which damage is being accumulated iin service)
2. a valid quantative comparison of structural life before and after Dpatch installation, and
3. any convenient additional diagnostic information.

The Dosimeter is a key component of the process to design and install the most effective patch possible. In order to provide this function the dosimeter must meet several goals:

- The dosimeter should be simple to install/dis-install on a widest practical variety of aircraft and locations.
- The dosimeter should measure high frequency strain and temperature while the aircraft is operational.
- The dosimeter should operate autonomously.
- The dosimeter should be "affordable".

To meet these goals the approach includes:

- Building the dosimeter from commercial off-the-shelf parts. This enables the construction of a dosimeter that is low-cost yet small enough that it can be installed on most aircraft for most hcf locations.
- Packaging the sensors, processor, and battery separately. The dosimeter design allows for a generous cable run (up to 50 feet) between the sensors (dynamic strain and temperature) and the processing/storage unit. The dosimeter power source is a battery, so that the dosimeter can operate autonomously. The battery will be packaged separately so that each package (battery and processor/storage unit) can be as small as possible, allowing more latitude in dosimeter installation. Additionally, this allows the battery size to be adjusted in the future as requirements (operational times) changes.

- Utilizing state of the art programmable digital signal processor (DSP) computer chips. This allows the dosimeter to function autonomously, by waking up at regular intervals, and statusing the sensors for significant activity. Similarly, the dosimeter puts itself into a sleep-mode when sensor activity has been insignificant for some predetermined period of time. A C-programmable DSP also offers the advantage of configurability by downloading new programs via the dosimeter's serial port.

These concepts have been used for the preliminary design of a prototype dosimeter processing/storage unit.

This design has enough processing and memory capacity (a minimum of 1 mega-byte) to provide sufficient flexibility in computing the necessary design information for patch design. At present the procedure for using the dosimeter can be outlined as follows:

- **Installation** of the dosimeter. The sensors are bonded to the damage prone area. The dosimeter can be installed on an aircraft with, or without an existing HCF crack. Without is preferred since the patch can be most effective as a preventive measure.

  Once the sensors, dosimeter processor/storage unit, and battery are installed, the dosimeter is powered on. There will be status LEDs to indicate that the dosimeter is in a powered standby state, ready to begin data collection.

- **Data-Collection** is performed autonomously as the aircraft is operated throughout a 3-10 day period. The dosimeter takes a single sample of strain data from the sensor. If the sensor RMS levels are significant, the dosimeter powers up, and records data until sensor activity reduces to insignificant levels.

  The dosimeter records a time-history of strain each second, and processes the time-history for the remainder of each second. This method is valid as long as peak value strain detection is not important, which is the case with HCF cracking problems. Typically the structure is responding in a steady-state fashion. From each time-history, the RMS strain in certain 1/3 octave bands is computed, along with minimum, and maximum strain values and temperature. These data are saved in memory, along with "typical" and worst-case sample strain time-histories. If the dosimeter's memory should ever become full the dosimeter will power itself down so that the gathered data can be downloaded.

- **Data-Removal** is performed over a serial-port with a laptop computer. At this point the dosimeter can be removed from the aircraft, or left installed to verify that the newly designed durability patch meets its performance goals.

The maximum overall level is not expected to exceed 3000 micro strain peak. The frequency range of interest is from 44.7 Hz (the lower limit of the 50 Hz band) to 2239Hz (the upper limit of the 2000 Hz band). It is required that the dosimeter be mounted on the aircraft so as to remain in place after high g loading; damage to dosimeter and to aircraft is acceptable as a result of any high g loading. The dosimeter is designed such that it will not be intrusive on operations.

5 BONDED REPAIR

The technology base for application of bonded repairs to aircraft structure has achieved a threshold of maturity sufficient to support this effort. Structural repair materials, structural adhesives, surface preparation techniques, design methods, and installation processing and procedures are well established. There are many applications performing satisfactorily in service, many of which are for primary structure. Bonded repair technology is well documented.[6,7,8] One recommended design practice is that the patch match the extensional membrane stiffness of the baseline structure in order to avoid load attraction or shedding. Single sided repair results in eccentricity of load which induces bending stresses which must be accommodated.

The design concepts for patches used in bonded repair of primary structure are monolithic and laminated. Structural patch materials are aluminum, fiber metal laminate (FML, eg, GLARE, an aluminum and FG laminate), graphite fiber/epoxy prepreg, and boron fiber/epoxy prepreg.

Applications of bonded repair to primary structure is far more demanding than applications to secondary structure, where there are no significant safety of flight concerns. However, some aspects of bonding are exacting.

6 DAMPING

Viscoelastic vibration damping technology has achieved a level of maturity sufficient to support this effort.[3] The stand off damping treatment configuration has been established as possessing very high modal damping performance, high weight efficiency, and significantly less dependent on temperature. Conventional constrained layer damping is flying in service in air flow on external surfaces, some with an edge sealant and some with their perimeter adhesively bonded. The highest practical levels of damping will be used; this will enhance the life of the repaired skin, and will also enhance the life of adjacent bays of skin and substructure. This approach is judged to be appropriate in the context of demonstrated opportunity for improvement in durability with respect to nuisance cracking. Often the intrinsic damping is low; this fact makes the structure more susceptible to resonant high cycle fatigue cracking. This fact also increases the benefits of damping because rms stress levels are highly dependent on modal damping. The dynamic magnification factor is inversely proportional to the square root of modal damping. The modal strain energy (MSE) has been established as the proper approach to calculate modal damping.[9,10,11]

7 CONCEPTS

The presumption here is that the Durability Patch will be a bonded repair; advantages and disadvantages of bonded and mechanically fastened repair are well established and documented[6,8] and will not be repeated here. A further presumption is that the installation of the bonded repair will be on the flight line at an operational base; this is considered to be somewhere between very challenging and unrealistic/impossible by many experts in bonded repair of primary structure. It is noted that the direct economic and technical consequences of extensive disbonding of a Durability Patch is minor and that this type of repair is a very low profile application. This situation may be used to good advantage in order to maximize benefits.

The fundamental purpose of the Durability Patch Program is to establish a repair technique for secondary structure (or other lightly loaded structure) which has been cracked due to sonic fatigue. The repair consists of restoration of structural integrity, which implies both static load carrying ability and life considerations. Very importantly, the Dpatch must offer an attractive option (relative to conventional techniques) to the potential user, or it will not be accepted. This means that it must be simple to install, require no more manhours than conventional repair, require no more clock time, no more requirements for aircraft environment, environmentally safe, etc. It must result in net cost savings with no adverse effects.

The following design philosophy points summarizes these factors:

- RESTORE STATIC CAPABILITY
- ENHANCE LIFE
- MINIMUM QUALITY ASSURANCE/INSPECTION
- COST SAVINGS
- EASE OF INSTALLATION

- AERODYNAMICALLY SMOOTH

To amplify on each of these points, the static capability of the structure must be restored. It is well known in bonded repair that the extensional membrane stiffness of the original skin should be closely matched to avoid load attraction and load shedding. Of course this is true only if the structure carries significant stress. Regardless, the repaired structure must be capable of carrying any applied loads. Since the existence of nuisance cracking demonstrates the opportunity for improvement in durability, the local flexural stiffness should be enhanced in order to better withstand loads.

The life of a properly designed and installed bonded repair will exceed the life of the undamaged baseline structure, although that is known to offer opportunities for improvement. The Dpatch must withstand moisture for decades, must reduce stress intensity and consequent crack growth rate, should reduce static stresses, and must reduce dynamic stresses. These points suggest no stress concentrations or hard points, vibration damping, and high tolerance of large disbonds/porosity.

In order to minimize costs, there should be a minimum of quality assurance and in service inspection. Measurement and recording of temperatures during cure, and a visual and coin tap afterward are probably the only requirements. No scheduled in service inspection is being considered.

In the interests of aerodynamic smoothness, the maximum thickness will be 1/8 inch, which is negligible with respect to the boundary layer on the aft 80 percent of any surface; also, a beveled edge with a en to one slope will be incorporated.

The context may be summarized by the following list of points:

- Need for restoration of structural integrity of cracked secondary structure
- Demonstrated opportunity for improvement in durability
- Bonding installation on flight line by inexperienced personnel
- Minor direct consequences of large disbonds
- Opportunity for developing bonding personnel/service experience

Concepts for different aspects of the Durability Patch are listed:

- Prep of crack: stop drill, scarf, seal
- Design: 1-, 2-sided, monolithic, laminated, sandwich (edge: beveled, square, etc)
- Planform: oval, rectangular, fingered
- Perimeter: sealant, integrally tapered core, beveled
- Structural materials: aluminum (2024, 1100, etc; sheet; foil, ribbon, wire), FML, GLARE, fiberglass/epoxy (E or S glass), graphite/epoxy, boron/epoxy, quartz/epoxy, etc.
- Sandwich Core materials: syntactic foam, structural adhesive, etc
- Life enhancement: reduce crack growth rate (choice of structural material, laminations, etc), beef-up(ie, reduce static stresses), damping, etc.(ie, reduce vibratory stresses)
- Damping: stand off (spaced) constrained layer; structural adhesive perimeter

- Damping Stand Off Layer (grooved): syntactic foam, structural adhesive, etc
- Structural Adhesives: film: FM73; paste: etc.
- VEM: PSA, bonded, co-cured,

There are many advantages to a sandwich repair configuration: increased flexural stiffness reduces the eccentricity of the load path due to in-plane loading; reduced patch bending; reduced bending of the original skin, reduced peel stresses, reduced stress intensity at the crack tip - increased flexural stiffness reduces the curvature of the original skin at the crack due to vibration, reduced stresses skin, patch, adhesive, reduced stress intensity factor for vibration.

A sandwich Durability Patch design concept has been selected for further study as well as others. It will be further evaluated for all criteria and parameters. Near the center, there is an elliptical sandwich region which functions as a repair; the outer face sheet and the core of the sandwich is extended in all directions to form a rectangular overlay of standoff damping.

Layer 6 is a structural adhesive film next to the original, cracked skin. Layers 5A, 5B, 5C,...., are graduated ellipses; collectively they serve as one face sheet of a sandwich repair area. Successive layers are slightly smaller to provide a gradual taper in thickness for aerodynamic smoothness and also for a consequent gradual change in stiffness. The total thickness provides one half or slightly more of the extensional stiffness of the original skin. Layer 3 is a rectangle of standoff layer having a beveled perimeter; it is stiff in shear and soft in flexure. Layer 2 consists of a rectangle of viscoelastic damping material with an elliptical insert of structural adhesive. The external layer (1A) is a prepreg and is rectangular with a generous radius in the corners; it extends beyond all other layers of the patch where its perimeter is adhesively bonded to the original skin. Successive graduated layers of prepreg (1B, 1C,....) contact only vem at their perimeter. The elliptical region of layer 1 (face sheet) and 3 (core) which shadows layer 5 (face sheet) serves as a symmetric sandwich static repair. The remaining region of layer 1 and 3 serves as the constraining layer and stand off layer respectively of a damping treatment. Substantial material and installation cost savings result from this highly integrated, multifunctional design concept.

The prepreg layers have the advantages of conformability, ease of installation, and aerodynamic smoothness. It also avoids a secondary operation of sealing the edges and installing an aerodynamic ramp. At this juncture, the only advantage of aluminum appears to be the possible residual compressive stress from cure and the consequent very low crack growth rates. A disadvantage of aluminum is the lack of conformability for significant thicknesses and compound curvatures. These aspects will be evaluated in the near future.

The area to be covered by a Dpatch installation would be two bays of skin covering the fastener row between the bays and almost to the perimeter fastener row, leaving room for the sealant line of a vacuum bag. Procedures will be investigated to accommodate fasteners which must be removed.

8 DESIGN

A sandwich design concept is described above. Finite element analysis will be performed to arrive at values for stresses for various parts of the repaired structure including in the original skin at the edge of the patch, the patch, the adhesive, etc., for a variety of loading conditions. The major design considerations are static stresses, vibratory stresses, and stress intensity and their effect on static strength, low cycle fatigue life, high cycle fatigue life, and crack growth rate. FEA stresses will be compared with the strength and fatigue allowables for the various materials constituting the Durability Patch. Stress intensity factors will also be investigated using finite elements for input to calculations to crack growth rates. The modal strain energy[9][10][11] will be used for calculations of modal damping. Viscoelastic damping materials will be selected to provide the highest practical damping in the fundamental mode at the service temperature. The maximum damping will protect adjoining bays of structure somewhat.

9 DISCUSSION

The Durability Patch program has major payoff in cost avoidance savings in maintenance and repair of sonic fatigue cracking. Extensive annoyance cracking of secondary structure occurs in service. The consequences of this cracking are large cost of repair and maintenance and reduced operational readiness; there are no ramifications with respect to safety of flight. When annoyance cracking occurs at the same structural location on a substantial portion of the fleet at a small fraction of the intended service life, an opportunity for improvement in durability has been amply demonstrated. Aging aircraft are subject to even more annoyance cracking.

A major benefit of the Dpatch is the minimal potential for additional damage because repairs are made in-situ which minimizes handling damage. The DPP has additional payoff beyond the program and repair in that service experience for bonded repair and a pool of personnel skills will be developed. Furthermore, experience is provided for future applications of micro data collectors analyzers loggers, eg, health monitoring.

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